

EXPERIMENTAL INVESTIGATION OF AN INTEGRATED STRUT-ROCKET / SCRAMJET OPERATING AT MACH 4.0 AND 6.5 CONDITIONS

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ABSTRACT

A series of tests were conducted to investigate RBCC performance at ramjet and scramjet conditions. The hardware consisted of a linear strut-rocket manufactured by Aerojet and a dual-mode scramjet combustor. The hardware was tested at NASA Langley Research Center in the Direct Connect Supersonic Combustion Test Facility at Mach 4.0 and 6.5 simulated flight conditions.

INTRODUCTION

A rocket-based combined-cycle (RBCC) engine is unique in that it combines rocket and airbreathing components into a single propulsion unit. There are many variants of the RBCC, but perhaps the simplest are the ejector ramjet and ejector scramjet. Modern ejector scramjet RBCCs are basically modular, or 2-D, scramjet ducts with several rocket ejectors mounted in the bases of fuel injector struts or in steps along the side-walls of the duct. A number of these designs are currently being tested under NASA - Marshall Space Flight Center's ARTT program.

The modes of operation of the RBCC vary as the vehicle accelerates through the atmosphere (and into space for a launch vehicle). In general, the ejector scramjet's modes are from air-augmented rocket through Mach 3, ramjet and scramjet through Mach 8 to 15, followed by a conventional rocket mode with a very large area ratio to orbit. It is the ramjet and scramjet modes that were the focus of this study.

BACKGROUND

For an accelerator class vehicle, which includes launch vehicles, it can be shown that both specific impulse (I_{sp}) and thrust to weight ratio (T/W) are of great importance to the overall performance of the vehicle. Starting with a simple free body diagram of a vehicle with a horizontal trajectory and small, off-axis angles, summing the forces in the direction of travel yields $F = T - D$. Here, T is the thrust and D is the drag.

For an accelerator, the change in velocity per unit change in mass is critical. dV/dm can be derived as follows:

$$\frac{dV}{dm} = \frac{dV/dt}{dm/dt} = \frac{a}{\dot{m}}$$

$$\frac{dV}{dm} = \frac{F}{\dot{m}} = \frac{T - D}{\dot{m}}$$

$$\frac{dV}{dm} = \frac{I_{sp} g T - D}{m T}$$

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$$\frac{dV}{dm} = \frac{I_{sp}g}{m} \frac{T/W - \frac{L/W}{L/D}}{T/W}, \quad (1)$$

where L is lift and W is weight. Assuming T/W , L/W , L/D and I_{sp} are averaged values over the velocity range of interest, integrating (1) yields

$$\Delta V = -I_{sp}g \frac{T/W - \frac{L/W}{L/D}}{T/W} \ln(1 - \zeta), \quad (2)$$

where ζ is the propellant mass fraction.

By choosing a ΔV and ζ , a curve can be generated from equation (2) as shown in Figure 1. Based on the operational range of a ramjet / scramjet (typically $3 < M < 8$), the curve shown corresponds to a ΔV of 5000 ft/sec, a L/D of 2.5, and a L/W of 1.0. A ζ was chosen such that the curve passes through the T/W and I_{sp} values of H_2/O_2 rockets. Note that ramjet / scramjet powered vehicles lie on the vertical portion of this curve where $T \approx D$.

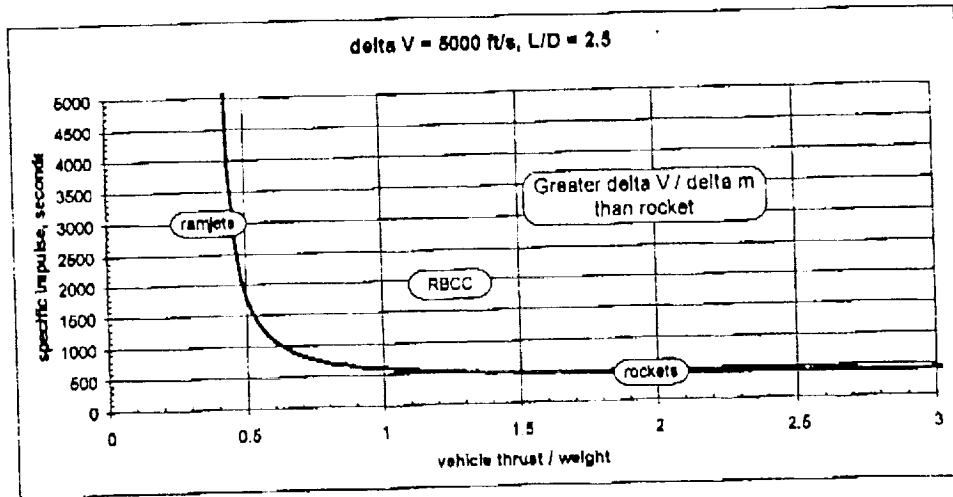


Figure 1. -

From this simple approximation, vehicle / propulsion systems located above and to the right of the curve would have better overall performance (lower ζ) than the systems that are currently available. In the case of a RBCC, this relation suggests that operating the primary rocket ejectors in the ramjet / scramjet mode will indeed reduce I_{sp} , but overall performance could be significantly greater due to the additional T/W .

OBJECTIVE

The main objective of this research was to test an integrated strut-rocket / scramjet and collect data both with and without rocket ejector operation. In so doing, the information gathered should provide a better understanding of the dynamics of the flow when operating in the ramjet and scramjet modes as well as a clue to any advantages or disadvantages to operating the rocket while in these modes. Other objectives of this study were to develop a means to measure the thrust produced during the runs directly and to generate an early RBCC database.

APPARATUS

The experimental apparatus consisted of two distinct systems, the facility heater and the linear, strut rocket. This hardware is subsequently described along with an explanation of the assembled sections that make up the scramjet duct. The reader is referred to Figure 2 for a schematic of the apparatus.

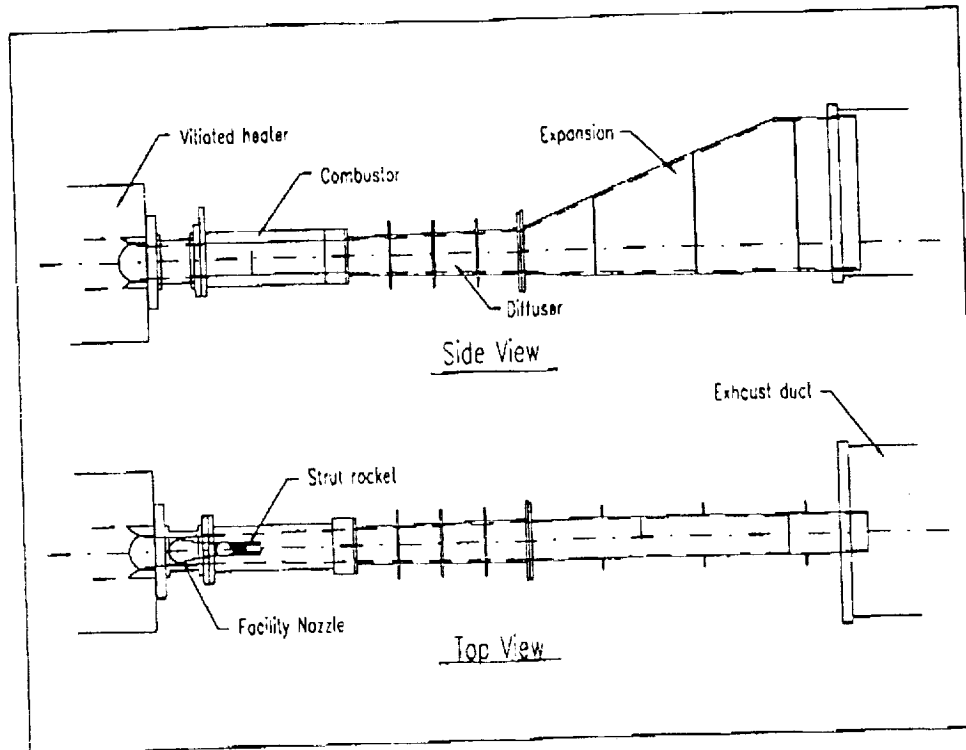


Figure 2. - Schematic of assembled apparatus.

FACILITY HEATER

The hardware is being tested in the Direct Connect Supersonic Combustion Test Facility at NASA Langley Research Center. The facility's hydrogen/ oxygen/ air vitiated heater is capable of simulating flight total enthalpies up to Mach 7.5. For this study, a Mach 2.6 centerbody, facility nozzle is being used to mate the heater to the scramjet combustor. The nozzle section transitions the heater flow from approximately 9 in. (20 cm) in diameter to a rectangular flow that is 5.79 in. (14.7 cm) vertical by 4.98 in. (12.6 cm) horizontal. The nozzle also serves as the leading edge of the strut that houses the rocket ejector. The throat area of the nozzle is approximately 5.7 in² (37 cm²).

STRUT ROCKET

The gaseous hydrogen/ oxygen linear, strut rocket was manufactured and previously tested by Aerojet Propulsion Company.⁴ It is made of nickel and was constructed using platelet technology. The assembly contains three individual, rectangular (2-D) rockets separated by structural stiffeners. The rocket is both water cooled and hydrogen film cooled. The exit area is 3.50 in² (22.6 cm²) and the exit area ratio is nominally 12.4. The maximum allowable chamber pressure is approximately 2000 psi (14 MPa). The overall external dimensions are 6.70 in. (17.0 cm) high by 1.27 in. (3.24 cm) wide by 4.76 in. (12.1 cm) long. Note that the rocket was not equipped with an ignitor, nor can one presently be installed. Therefore, a 20% silane (SiH₄) solution is used for "backlighting" the rocket.

SCRAMJET DUCT

The scramjet duct consists of three sections: a constant area combustor, a divergent section, and an expansion section, or exit nozzle. The initial length of the constant area combustor houses the rocket strut assembly, where the rocket is fastened vertically between two nickel plates. At the base of the plates are eight 1/8 in. (3.2 mm) diameter sonic fuel injectors that inject gaseous hydrogen tangential to the vitiated air flow. The total width of the rocket and the injector plates is 2.25 in. (5.71 cm), leaving a gap width of 1.37 in. (3.47 cm) on either side of the rocket assembly. Also, the copper combustor provides a reward facing step located downstream of the ejector as a possible fuel injector location. The overall length of the combustor section is 24.8 in. (63.0 cm) and the average wall thickness is approximately 1.9 in. (4.8 cm).

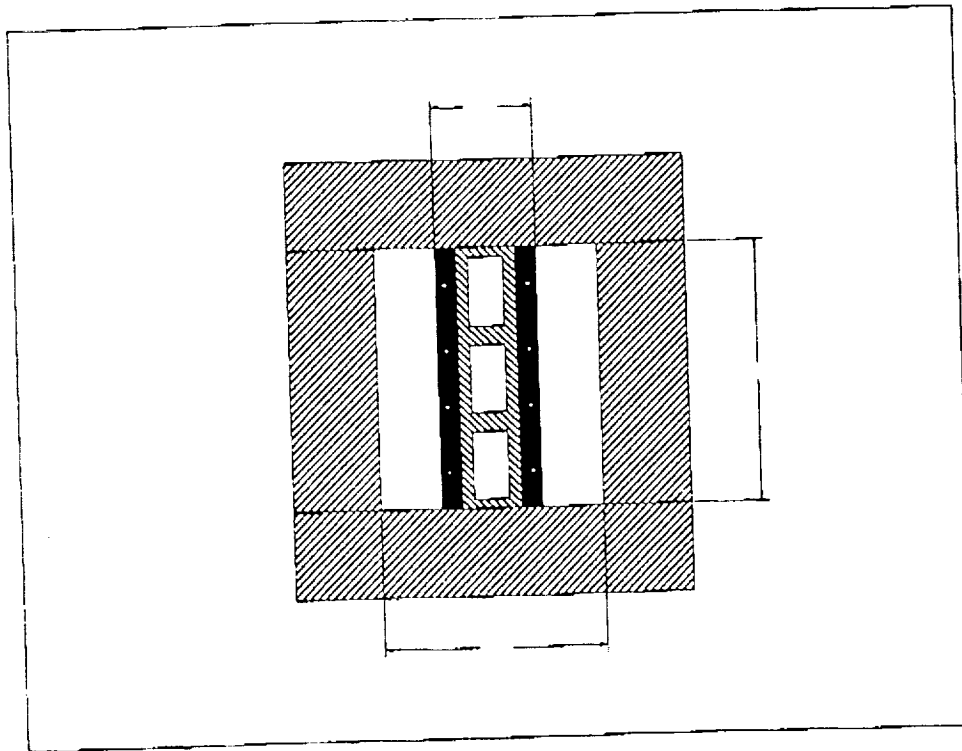


Figure . -

The divergent section is made of stainless steel. It expands on all four sides by approximately two degrees per side. Its length is 31.4 in. (79.9 cm) and the wall thickness is 0.5 in. (1.3 cm). The top surface of the pre-existing expansion section slopes upward at approximately 20 degrees. The axial length of the sloped section is 46.0 in. (117 cm). At the end of the expansion section is a constant area portion that fits freely within the exhaust duct flange. This duct then leads to the vacuum sphere. Figure 3 illustrates the area distribution of the duct and the corresponding axial length.

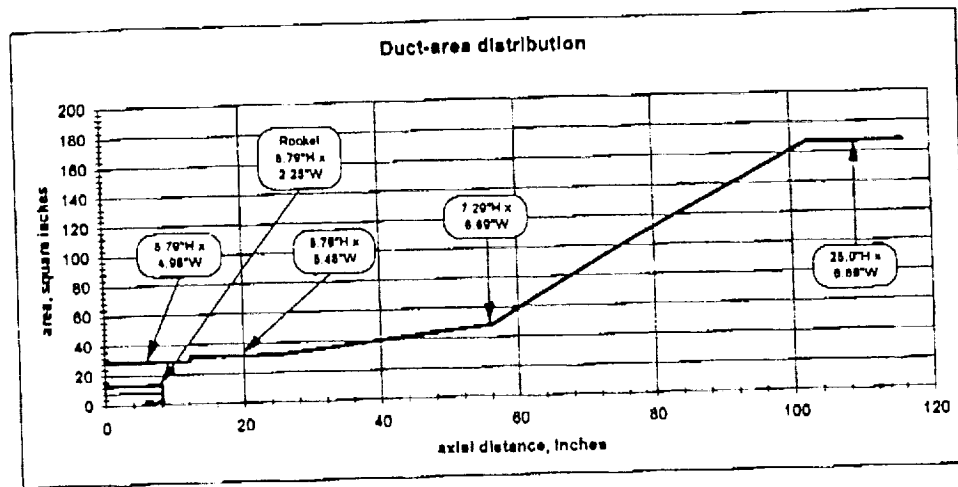


Figure 3. - Area distribution of duct.

INSTRUMENTATION

The facility heater, fuel injector, and strut rocket plumbing is equipped with instrumentation to calculate the mass flow rates of the supplied gases. Heater total pressure, total temperature, and rocket chamber pressure are also measured. Over 200 static pressure taps are located on the scramjet duct walls. In addition, a direct thrust measurement system, which is described in detail below, was designed for this experiment.

THRUST MEASUREMENT

Since the flowfield of an ejector type RBCC is very difficult to determine analytically, and interpreting the pressure integral from static wall pressure taps can be misleading, a direct thrust measurement system was developed for this experiment. Two 1 in. thick stainless steel plates with a slip-joint, o-ring seal are located between the facility nozzle section and the scramjet combustor section as shown in Figure 4. A tension/compression, strain-gage load cell rated to 3000 lb. is located between the plates. The plates provide a metric break so the net forces downstream of the facility nozzle are independent of the facility. Since the rocket and injector plates are fixed to the scramjet combustor section, their contribution is also measured. Two adjustable carriages were designed to suspend the scramjet hardware from large "H" beams with linear bearings. The pillow-block bearings allow the hardware to move axially. This arrangement also provided a means to align the individual pieces and simplified hardware assembly and disassembly.

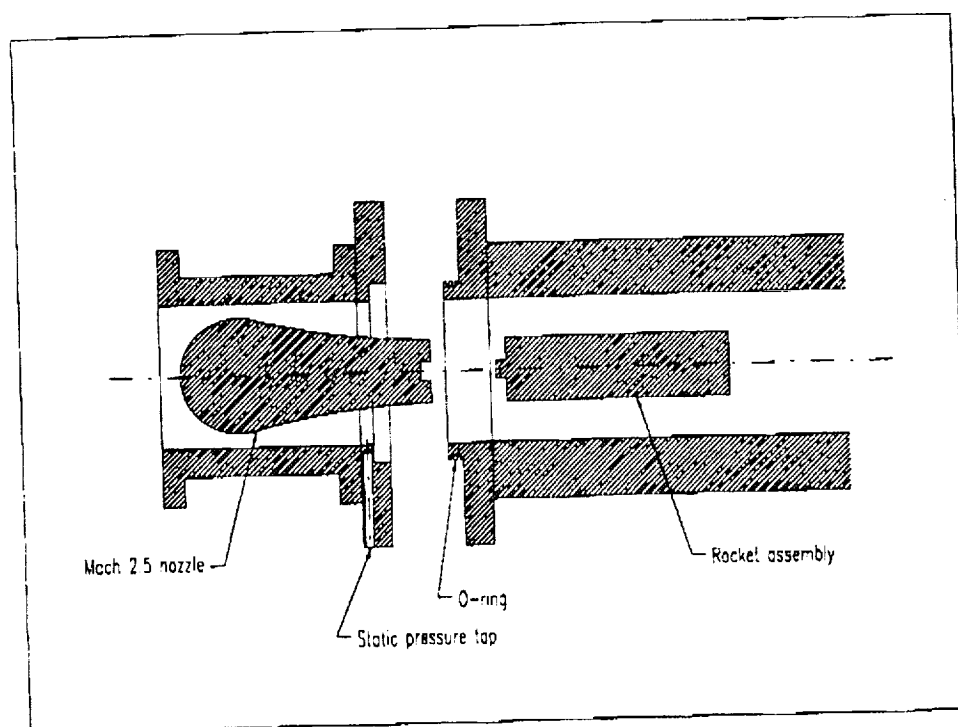


Figure 4. - Illustration of metric break for thrust measurement.

EXPERIMENTAL METHOD

FACILITY OPERATING CONDITIONS

The facility operating conditions chosen correspond to total temperatures at flight Mach numbers of 4.0 and 8.5. The total temperature values are 1600 °R and 3400 °R (890 K and 1900 K), respectively. The overall heater mass flow rates are such that an oxygen mole fraction of 20.95% is maintained. Since the effect of over and under-expansion of the rocket is of interest and the injected hydrogen mass flow rates were limited, a nominal heater exit pressure of 0.5 atm (50 kPa) was chosen. Table 1 lists the heater conditions associated with an exit pressure of 0.5 atm (50 kPa) at the chosen total temperatures.

M_∞	T_T °R	P_2 atm	P_T psia	mdt lbm/s	A_2 in ²	T_2 °R	F_2 lbf	U_2 ft/s	M_2	A_2 in ²	mdt _{O2} lbm/s
4.0	1600	0.500	132	9.62	5.67	743	1132	3400	2.51	15.8	2.31
8.5	3400	0.500	115	5.44	5.67	1880	996	5200	2.41	15.8	1.39

Table 1. - Heater operating conditions.

ROCKET OPERATING CONDITIONS

The key parameters for rocket operation are the oxidizer to fuel ratio (O/F), the chamber pressure (P_T), and the film-cooling fraction. The rocket was designed to run fuel-rich. This is appropriate for this RBCC application since the afterburning of the high temperature rocket exhaust gases is of interest. The injected O/F ratio for this experiment will vary from 0 to 8, O/F=8 being stoichiometric.

The amount of rocket film-cooling is characterized as a percentage of the total hydrogen flow into the rocket. A nominal value of 40%, also used during the Aerojet tests, was chosen for this experiment. However, some tests at lower O/F ratios may be performed later in the test program with less film-cooling. The rocket operating pressure is limited by two factors, both related to film-cooling. The first limitation is the hydrogen supply pressure. Due to the small film-cooling circuits, film-cooling flow rates greater than 0.14 lbm/s (0.068 kg/s) are not attainable. The second limitation is heat flux. As the rocket O/F ratio and chamber pressure increase, the heat flux also increases. Depending on the film-cooling fraction chosen, the amount of film-cooling required may exceed the amount supplied. The maximum allowable heat flux chosen for this study is 15 Btu/in²-s (27 kJ/cm²-s). This value is in agreement with the previous Aerojet tests.

Rocket conditions based on a chamber pressure of 300 psia (2.07 MPa) are presented in Table 2. Also, Figure 5 shows the operating range for the rocket with a film-cooling fraction of 40%. Unfortunately, even after the installation of a new high pressure hydrogen and oxygen system, the desired chamber pressures of 2000 psia achieved by Aerojet were unobtainable.

O/F	P _T psia	T _T °R	mdt lbm/s	A ₁ in ²	P ₂ atm	T ₂ °R	F _{2TOT} lbf	U ₂ kft/s	M ₂	A ₂ in ²	mdt _{H2} lbm/s	mdt _{fm} lbm/s
2.00	300	3620	0.328	0.318	0.149	1090	161	13.2	3.77	3.50	0.082	0.073
4.00	300	5420	0.350	0.318	0.197	2370	186	13.3	3.43	3.50	0.035	0.047
6.00	300	6110	0.382	0.318	0.251	3740	171	12.6	3.17	3.50	0.013	0.038

Table 2. - Rocket operating conditions at P_T=300 psia.

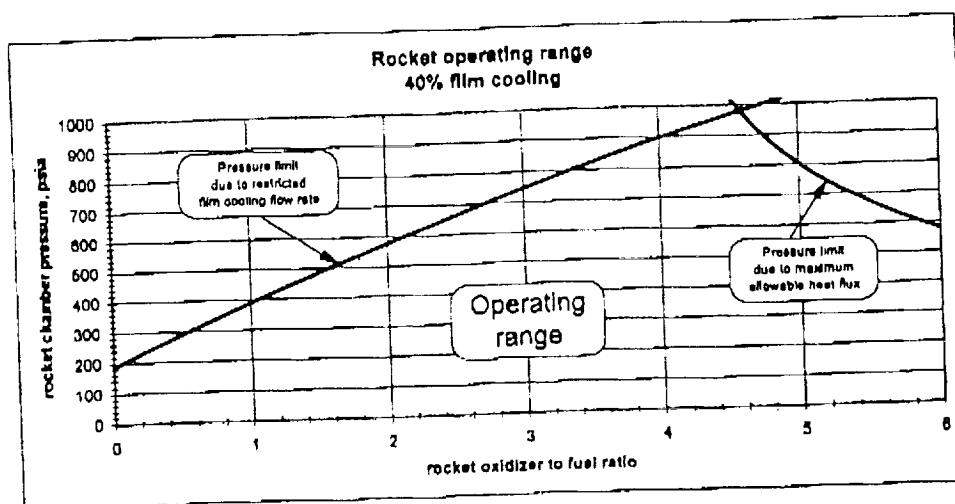


Figure 5. - Rocket operating range with 40% film cooling.

INJECTOR OPERATING CONDITIONS

As previously mentioned, the strut was equipped with eight tangential fuel injectors. The maximum hydrogen flow rate through these injectors was 0.17 lbm/s. This flow rate corresponds to fuel equivalence ratios of 0.6 at Mach 1600 °R and 0.9 at 3400 °R.

RESULTS

Nearly 100 "hot" runs were made with the vitiated heater operating and the RBCC hardware installed in the facility. Many of these runs were to check out the various systems and to establish a reliable ignition sequence for lighting and sustaining rocket combustion while the facility was running. The results that follow were obtained only from the data for runs that were designated as "good" based on

mole fraction of oxygen from the heater and relative closeness to the desired heater total temperatures and pressures.

To minimize the thermo-cycling of the hardware, the time at which the rocket and injector propellants were being fed overlapped. Therefore, each run actually satisfied three test conditions. First, while the heater was operating, the rocket propellant valves were opened to the desired settings. Then, with the rocket still firing, the hydrogen valve for the fuel injectors was opened. Next, the rocket propellants were shut off, leaving only the fuel injectors. And finally, the fuel injectors were turned off and only the heater was operating. The thrust data from this latter portion of the run was averaged with the other heater only thrust data and subsequently used as the "tare" thrust. It should be noted that the method for calculating thrust was simply to subtract the "tare" thrust from the raw, measured thrust. No attempt was made to calculate inlet drag and arrive at a net propulsive thrust. Thus, these thrust values cannot be compared to conventional rocket thrust data without some type of correction.

RAMJET SIMULATION

Table 3 below lists the run conditions and measured thrust and I_{sp} for the ramjet simulation at a heater total temperature of 1800 °R. The table is divided into three categories, rocket only, rocket + injector, and injector only. Note that for runs 99 through 106, the rocket was only supplied with hydrogen, no oxygen (O/F=0). All the ramjet simulation plots that follow were generated from this table of data. Φ , or ϕ , is the fuel equivalence ratio. The total fuel equivalence ratio is calculated based on the the sum of the theoretical excess hydrogen from the rocket, the film cooling hydrogen, and the injected hydrogen.

Rocket only 1600 °R

<i>run #</i>	<i>mdt inj</i>	<i>mdt rkt</i>	<i>Pt rkt</i>	<i>O/F rkt</i>	<i>PHI inj</i>	<i>PHI tot</i>	<i>F ref</i>	<i>Isp ref</i>
71	0	0.350	299	2.79	0	0.373	428	1220
72	0	0.412	339	3.83	0	0.314	471	1142
73	0	0.597	500	4.13	0	0.415	644	1079
76	0	0.429	337	5.26	0	0.225	358	835
77	0	0.457	372	2.05	0	0.597	583	1275
78	0	0.451	371	2.17	0	0.571	556	1233
79	0	0.437	367	2.08	0	0.581	561	1284
80	0	0.154	127	4.95	0	0.087	176	1145
81	0	0.649	535	3.80	0	0.482	700	1078
82	0	0.162	128	0.98	0	0.333	277	1706
83	0	0.141	121	1.53	0	0.229	256	1810
84	0	0.238	199	4.18	0	0.161	308	1293
99	0	0.158	80	0	0	0.579	461	2921
101	0	0.160	81	0	0	0.575	486	3039
106	0	0.149	77	0	0	0.552	426	2855

Rocket + Injector 1600 °R

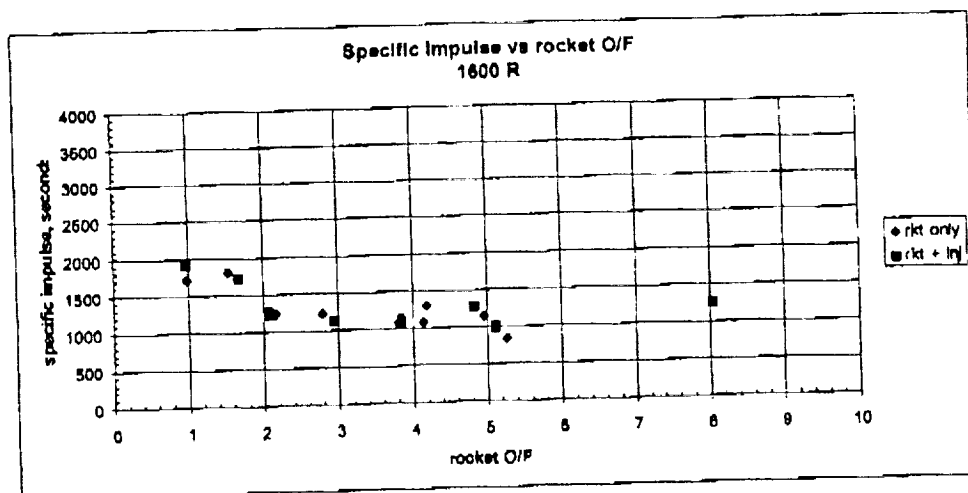
<i>run #</i>	<i>mdt inj</i>	<i>mdt rkt</i>	<i>Pt rkt</i>	<i>O/F rkt</i>	<i>PHI inj</i>	<i>PHI tot</i>	<i>F ref</i>	<i>Isp ref</i>
71	0.070	0.366	300	2.94	0.252	0.617	492	1131
76	0.098	0.421	338	5.12	0.360	0.587	518	998
78	0.065	0.443	362	2.11	0.231	0.808	624	1228
79	0.138	0.436	362	2.07	0.479	1.040	713	1242
80	0.121	0.150	126	4.82	0.420	0.504	349	1283
81	0.162	0.655	542	3.84	0.570	1.055	880	1077
82	0.097	0.161	126	0.97	0.334	0.655	495	1918
83	0.094	0.138	119	1.67	0.325	0.532	396	1708
84	0.130	0.213	161	8.03	0.446	0.495	428	1248
99	0.148	0.159	80	0	0.543	1.127	740	2419
101	0.074	0.160	81	0	0.264	0.834	678	2898
106	0.075	0.149	77	0	0.279	0.836	585	2615

Injector only 1600 °R

<i>run #</i>	<i>mdt inj</i>	<i>mdt rkt</i>	<i>Pt rkt</i>	<i>O/F rkt</i>	<i>PHI inj</i>	<i>PHI tot</i>	<i>F ref</i>	<i>Isp ref</i>
71	0.071	0	0	0	0.257	0.257	252	3522
76	0.098	0	0	0	0.359	0.359	300	3065
78	0.065	0	0	0	0.232	0.232	261	3997
79	0.138	0	0	0	0.482	0.482	466	3377
80	0.116	0	0	0	0.396	0.396	382	3295
81	0.160	0	0	0	0.561	0.561	526	3287
82	0.095	0	0	0	0.326	0.326	333	3483
83	0.095	0	0	0	0.323	0.323	328	3463
84	0.129	0	0	0	0.453	0.453	491	3823
99	0.142	0	0	0	0.518	0.518	463	3273
101	0.076	0	0	0	0.267	0.267	280	3704
106	0.075	0	0	0	0.280	0.280	247	3270

Table 3. - Data for ramjet simulation (1600 °R).

Looking first at specific impulse, Figure 8 is a plot of specific impulse versus rocket O/F for the rocket only and rocket + injector tests. The data points appear to follow a trend where lower rocket O/F values provide higher I_{sp} independent of rocket chamber pressure. This is somewhat expected since less oxygen is being used at the lower O/F ratios. What is not expected is that the I_{sp} is not effected by the additional fuel through the fuel injectors. One might initially predict that the I_{sp} would increase with the added fuel, but this is indeed not the case here. This is the first indication that when the fuel is injected while the rocket is in operation, the fuel does not burn efficiently.

Figure 6. - I_{sp} versus O/F (1600 °R).

Similarly, Figure 7 is a plot of specific impulse versus fuel equivalence ratio for the H_2 only rocket tests and the injector only tests. From this plot one can see that the I_{sp} with the H_2 fed rocket is less than that of the H_2 injectors. Considering that the hydrogen exiting the rocket is around Mach 3.5, it is much colder than the hydrogen exiting the injectors. Thus, the "supersonic injector" rockets are less efficient than the sonic injectors and produce less thrust per unit mass flow. Looking at the H_2 rocket + injector data, once again, the added fuel injectors do not help the situation.

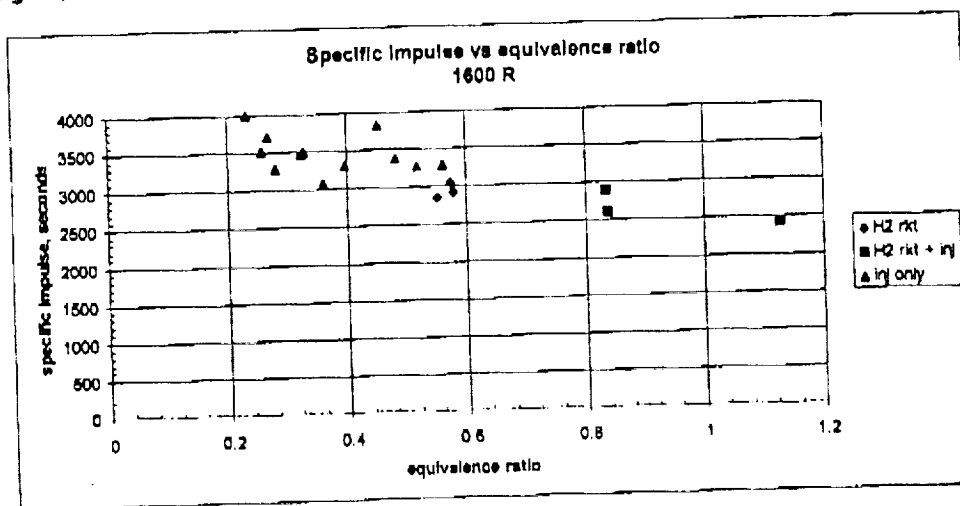


Figure 7. -

All the tests can be compared on a single plot of thrust versus propellant mass flow rate as in Figure 8. The slope of a line passing through the origin and the data point of interest is equivalent to the specific impulse. The hydrogen only data, solid symbols, lies along a fairly straight line with a larger slope, greater I_{sp} . The scatter in the H_2/O_2 rocket data is mostly due to the various rocket O/F values tested. Typically, lower rocket O/F yields higher I_{sp} as illustrated in Figure 6.

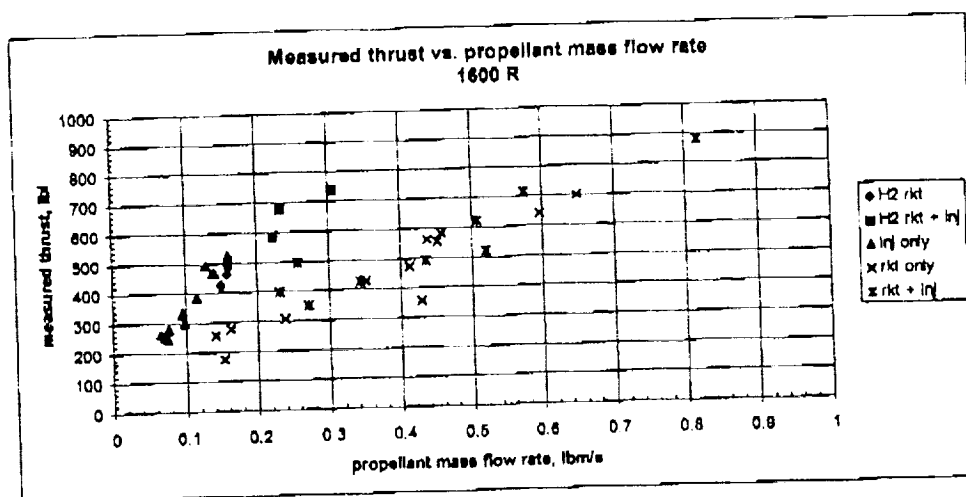


Figure 8. -

Previously, Figure 6 suggested that the rocket actually hinders the hydrogen from the fuel injectors from burning. In fact, a close examination of runs 80 and 84 (Table 4) reveals that the thrust produced by the injectors alone in these runs, is actually greater than the thrust produced with rocket + injectors, at the same injector mass flow rate. Apparently, the fuel from the injectors is being entrained into the rocket exhaust and "carried" downstream before it is able to mix with the air and burn. Under these circumstances, injecting the fuel from the side-walls would most likely have been a more efficient means of injection.

Run # 80

	<i>mdt inj</i>	<i>mdt rkt</i>	<i>Pt rkt</i>	<i>O/F rkt</i>	<i>PHI inj</i>	<i>PHI tot</i>	<i>F ref</i>	<i>Isp ref</i>
rocket	0	0.154	127	4.95	0	0.087	176	1145
rkt+inj	0.121	0.150	126	4.82	0.420	0.504	349	1283
injector	0.116	0	0	0	0.396	0.396	382	3295

Run # 84

	<i>mdt inj</i>	<i>mdt rkt</i>	<i>Pt rkt</i>	<i>O/F rkt</i>	<i>PHI inj</i>	<i>PHI tot</i>	<i>F ref</i>	<i>Isp ref</i>
rocket	0	0.238	199	4.18	0	0.161	308	1293
rkt+inj	0.130	0.213	161	8.03	0.446	0.495	428	1248
injector	0.129	0	0	0	0.453	0.453	491	3823

Table 4. -

Figure 9 is similar to Figure 1 except the vehicle drag and weight is not known so measured thrust is used instead of net thrust and T/W. Since the same hardware is being compared for all these tests, measured thrust is sufficient. As before, data points toward the upper right hand corner of the graph are the most desirable. Perhaps a better method of comparing the various tests is to choose those with the same total fuel equivalence ratio. Figure xxx is the same as Figure 9 but only includes tests points with total fuel equivalence ratios of around 0.53.

Clearly, for the hydrogen only tests (airbreather), the higher fuel equivalence ratios are the most desirable. These conditions produce greater thrust and the specific impulse is held pretty well constant to a point. In the case of the integrated H_2/O_2 rocket / ramjet, the specific impulse was around 1300 seconds for the range of rocket chamber pressures tested. Assuming this trend would continue for rocket operation at higher chamber pressures, the integrated rocket / ramjet may actually have better performance due to the enormous amount of thrust capability even though the I_{sp} is considerably less than

the H_2 only tests. Note, however, that this assumption depends heavily on the vehicle drag and weight. In addition, the specific impulse of the integrated system could be improved by injecting the additional fuel from the side-walls instead of the strut.

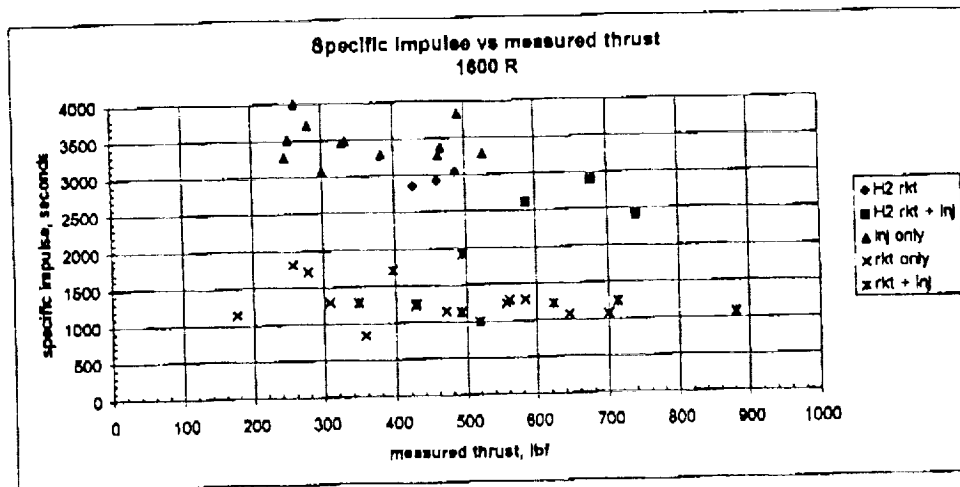


Figure 9. -

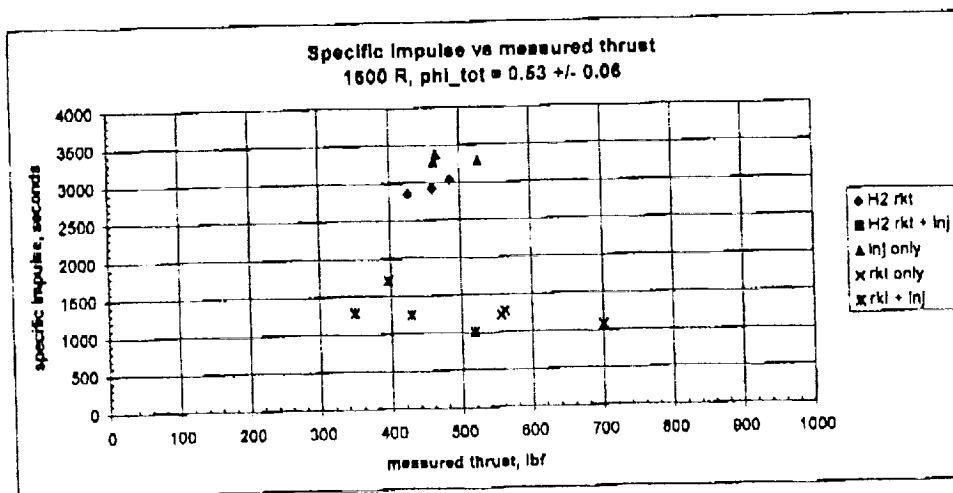


Figure 10. -

SCRAMJET SIMULATION

Table 5 lists the run conditions and measured thrust and I_{sp} for the scramjet simulation at a heater total temperature of 3400 °R. Again, the table is divided into three categories, rocket only, rocket + injector, and injector only. At this heater temperature it was very difficult to establish and maintain combustion of the hydrogen from the fuel injectors unless the rocket was in operation. Since it was desirable to reduce the run time on the hardware anyway, the injector only tests were discontinued after run 90 but resumed during the H_2 rocket tests. It was found that the injected hydrogen only burned if the fuel equivalence ratio was greater than 0.6. Tests where the fuel did not burn are not included in the table. Also, none of the hydrogen only rocket tests burned and were therefore also left off the table.

Rocket only 3400 °R

	<i>mdt inj</i>	<i>mdt rkt</i>	<i>Pt rkt</i>	<i>O/F rkt</i>	<i>PHI inj</i>	<i>PHI tot</i>	<i>F ref</i>	<i>Isp ref</i>
86	0	0.419	339	3.71	0	0.487	260	620
87	0	0.419	335	3.97	0	0.432	266	635
88	0	0.414	337	3.85	0	0.451	280	675
89	0	0.402	322	1.92	0	0.814	353	879
90	0	0.424	334	4.94	0	0.350	252	595
91	0	0.371	313	4.71	0	0.331	280	753
92	0	0.112	99	3.93	0	0.123	179	1604
93	0	0.630	501	4.73	0	0.559	479	760
94	0	0.688	552	3.66	0	0.813	456	663
95	0	0.725	544	6.72	0	0.370	450	620
96	0	0.194	150	1.56	0	0.458	147	757
97	0	0.284	209	7.90	0	0.108	127	447
98	0	0.163	129	3.94	0	0.174	122	747

Rocket + Injector 3400 °R

	<i>mdt inj</i>	<i>mdt rkt</i>	<i>Pt rkt</i>	<i>O/F rkt</i>	<i>PHI inj</i>	<i>PHI tot</i>	<i>F ref</i>	<i>Isp ref</i>
86	0.096	0.414	339	3.65	0.517	1.011	423	830
87	0.163	0.414	336	3.91	0.831	1.268	497	862
88	0.050	0.405	338	3.69	0.257	0.716	396	871
89	0.019	0.401	321	1.88	0.100	0.924	480	1141
90	0.123	0.438	338	5.12	0.652	0.991	440	783
91	0.123	0.382	318	4.90	0.655	0.972	537	1062
92	0.117	0.111	97	3.90	0.608	0.730	382	1675
93	0.037	0.638	512	4.89	0.204	0.748	553	819
94	0.039	0.688	562	3.64	0.211	1.028	562	773
95	0.081	0.716	553	6.65	0.429	0.797	592	743
96	0.126	0.193	149	1.55	0.668	1.132	296	926
97	0.170	0.286	209	7.82	0.888	0.998	360	791
98	0.174	0.158	124	4.17	0.906	1.063	365	1097

Injector only 3400 °R

	<i>mdt inj</i>	<i>mdt rkt</i>	<i>Pt rkt</i>	<i>O/F rkt</i>	<i>PHI inj</i>	<i>PHI tot</i>	<i>F ref</i>	<i>Isp ref</i>
87	0.164	0	0	0	0.833	0.833	326	1991
90	0.129	0	0	0	0.690	0.690	280	2177
111	0.160	0	0	0	0.826	0.826	309	1924

Table 5. - Data for scramjet simulation (3400 °R).

The hydrogen's unwillingness to burn was not a complete loss. It supported a potential benefit conceived at the beginning of this study that the rocket could also act as a flameholder and/or pilot in the airbreathing mode. Running in this combined mode could, therefore, reduce the need for complicated injection schemes.

There is some question with regards to the thrust measurement of run 92. In all of the other runs, the thrust is proportional to the rocket chamber pressure in the rocket only tests. At this time, no one rational reason for this anomaly has been pinpointed. One possibility is that offset is within the uncertainty of the calculation since the mass flow rates and rocket chamber pressure are much lower in this run than any of the others. For the time being, the data for run 92 has been included in the table, but is not included in the plots that follow.

In Figure 10, the rocket + injector tests performed slightly better than the rocket only tests. Recall that for the ramjet conditions the injector did not significantly improve rocket only performance. Also, the average I_{sp} is less than that for the ramjet tests as would be expected. This was also the case for the injector only tests as indicated in Figures 11 and 12.

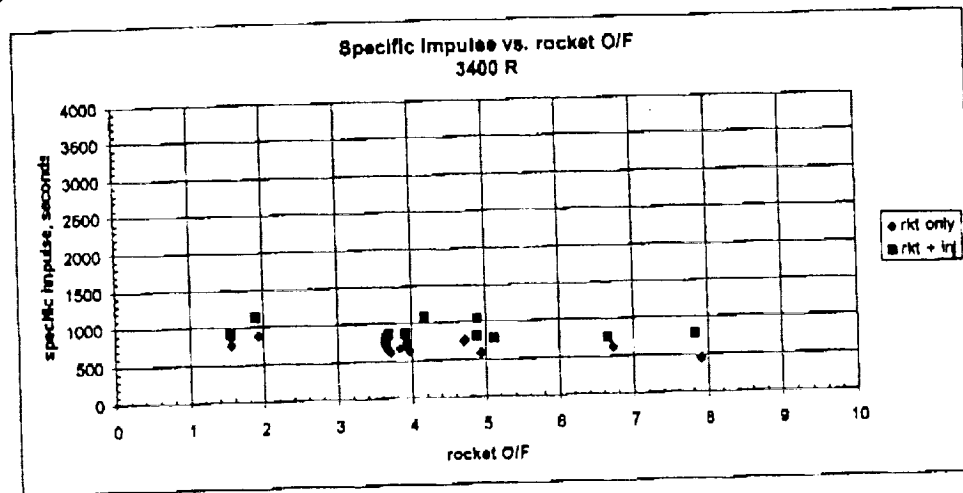


Figure 10. -

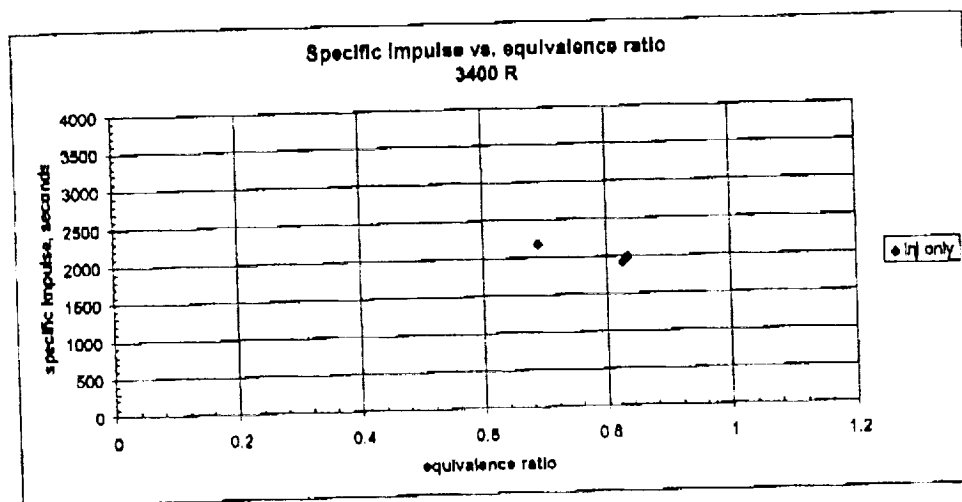


Figure 11. -

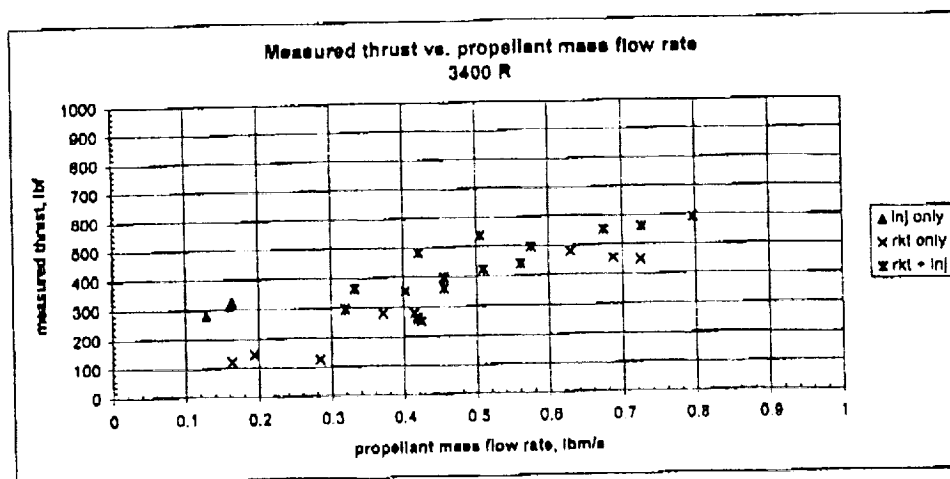


Figure 12. -

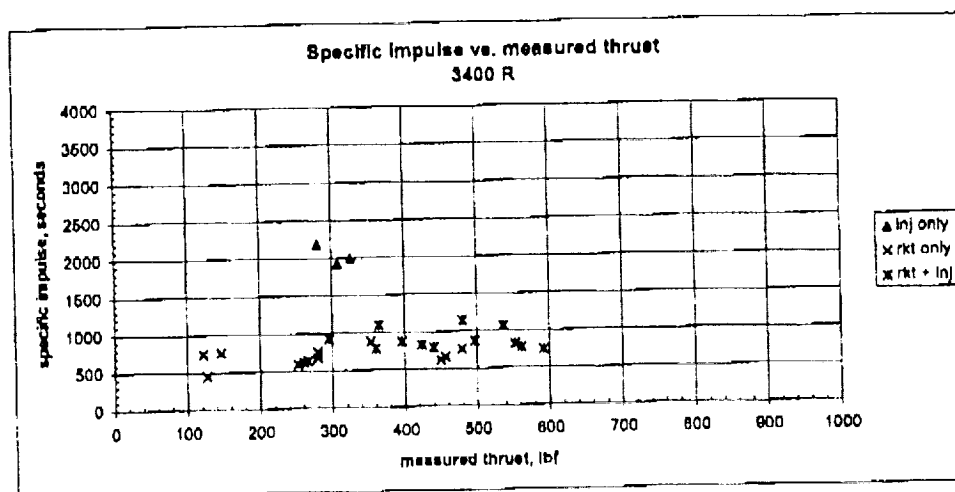


Figure 13. -

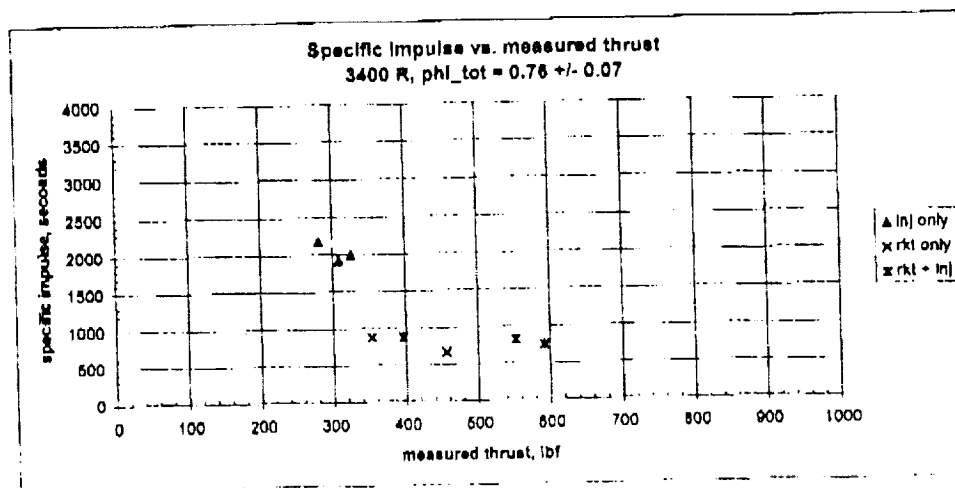


Figure 14. -

rocket / injector interaction

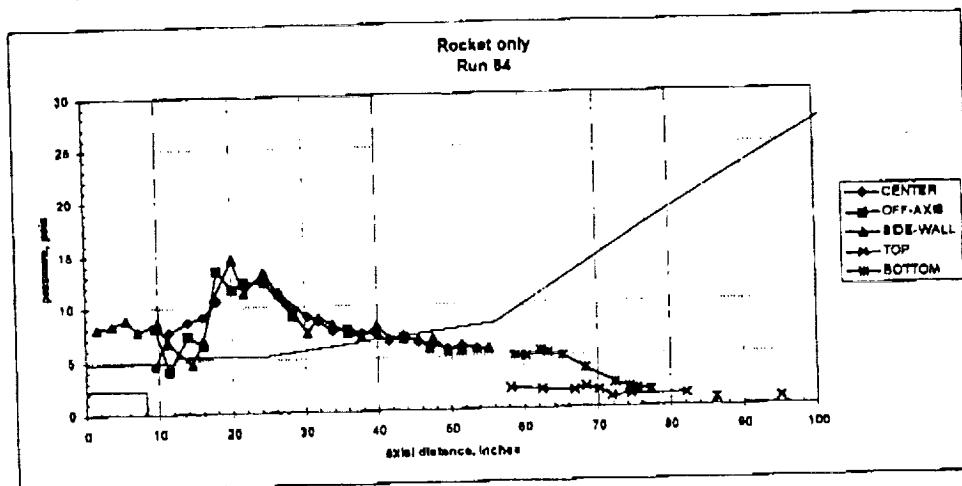


Figure 14a. -

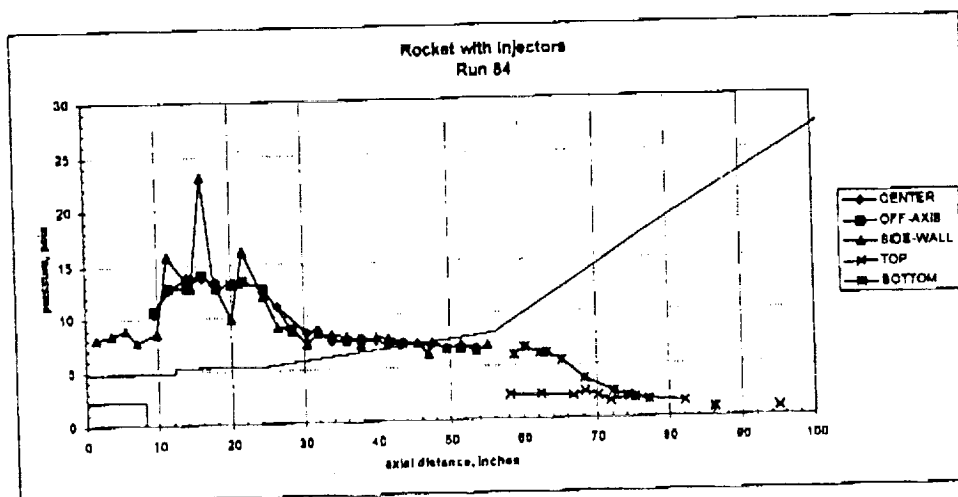


Figure 14b. -

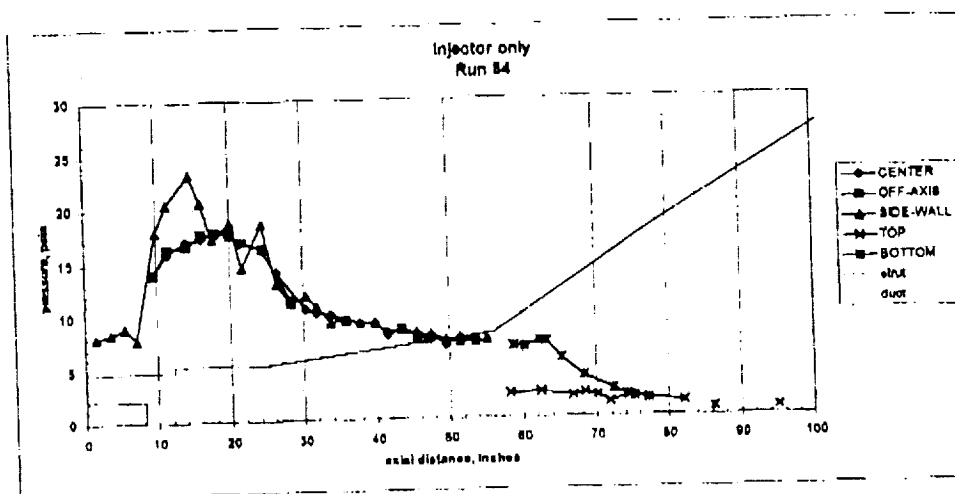


Figure 14c. -

thrust

summary and conclusions